

## NOZZLE CONTOUR DESIGN FOR SPACE PROPULSION MODULE

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### ABSTRACT

*The Earth based Trojan asteroid, later named as 2010TK7, was discovered by the WISE (Wide field Infrared Survey Explorer) space telescope in the year 2010. This paper highlights the design of the propulsion system module which provides sufficient thrust for onward journey to the Trojan 2010TK7 after the injection of the spacecraft from the launch vehicle. The propulsion module after separated from the reentry module will hit the Trojan surface for creating a particle cloud and thereby assisting the sample collection. In this process, the propulsion module gets self-destructed. The propulsion module and reentry module are kept inside the heat shield, and hence it is protected from all aerodynamic loads in atmospheric flight regime of the launch vehicle. The propulsion system design of the finalized configuration is followed by configuration design, engine design, and nozzle counter is developed by using method of characteristics (MOC) and analyzed using CFD++ software. Finally propellant quantity estimation will be performed in standard practice. The mission detailed in this work intends to perform a soft landing on the object, collecting some sample material and safely return it back to Earth for further scientific studies.*

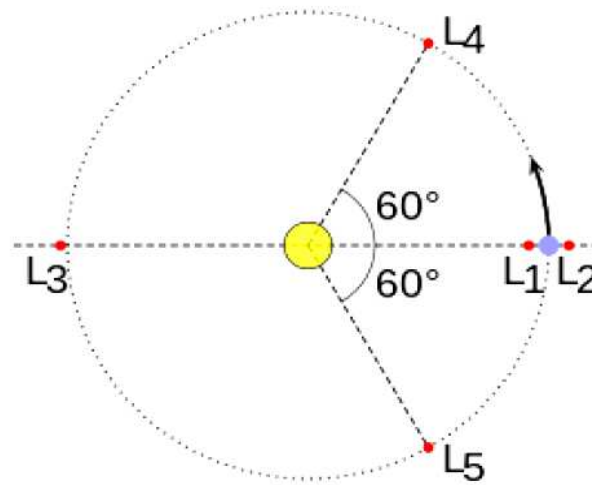
**KEYWORDS:** Spacecraft, Propulsion Module, Nozzle Counter & Asteroid 2010TK7

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### INTRODUCTION

The earth based Trojan asteroid, later named as 2010TK7 [1], was discovered by the WISE (Widefield Infrared Survey Explorer) space telescope in the year 2010. The detailed orbit analysis carried out later identified it as an object sharing the earth's orbit and revolving around the sun at the L4 Lagrangian point of earth. Though many Trojans were identified for the outer planets, 2010TK7 became the first earth based Trojan to be identified and studied. Most Trojans are expected to be in the L4 or L5 points of earth. Hence they are most likely to appear in the daytime sky and hence are extremely difficult to be identified and observed from earth's surface. The study of this object can not only through light into the origin of earth and the solar system, but also can be of huge socio-economical significance. Though the object is comparatively nearer to earth, it is hard to do any study using earth based or space telescopes due to its small size and rather odd orbital parameters. Detailed analysis of the orbital parameters of the target object revealed that the mission would be characterized by the very large delta-V requirement. The mission detailed in this document intends to perform a soft landing on the object, collecting some sample material and safely return it back to earth for further scientific studies. This paper highlights the design of the propulsion system module which provides sufficient thrust for onward journey to the Trojan 2010TK7 [1] after the injection of the spacecraft from the launch vehicle. Only one earth based Trojan has been discovered to date owing to the difficulty in locating such small bodies, orbiting sun at the Lagrangian points,

using the present-day technologies.



**Figure 1: Location of Lagrangian Points**

2010TK7 was discovered by WISE mission in 2010 and present day understanding of the object is limited to the orbital parameter details. No clue to shape, accurate size, composition, texture etc., are known till date. It is also noteworthy that no space agency has announced a mission to this object till date. Objects like 2010TK7 are either formed from the left over of the parent planet forming materials or could be a captured asteroid. Only close up studies and sample return missions can through light on the true origin of 2010TK7. In any case the data recovered may not only be scientifically exciting but may have many economical/commercial aspects too. The asteroid bodies are also formed from the same mix up of materials from which the planets were formed and hence should have identical composition.

## CONFIGURATION DESIGN

The design involves various configurations, a study was carried and their merits and demerits are detailed in the preceding section.

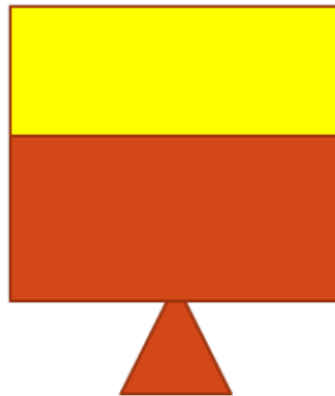
### Single Spacecraft Configuration

The spacecraft has to propel itself from earth to target object by performing a perform soft landing for sample collection. Upon sample collection, it has to takeoff from target and return to EPO. The module shall the attempt reentry. But this idea seemed to have the most complexities in terms of design and development. The energy requirement also seemed to be more. Due to the complexities involved and due to the large energy requirement, this configuration was dropped.

### Spacecraft with Separable Return Module

The spacecraft [4] would land and collect the sample. The return module will eject itself from the spacecraft and return to earth. Soft landing is heavily complex owing to the very low gravity of target and uncertainty in its shape and size. The spacecraft would be heavy forcing the designers to go for lunar gravity assist which is too complex for our levels. This configuration seemed feasible but was found to be too complex for the scope of this course project. Figure 2 gives a glimpse of the configuration. The yellow portion is the reentry module while the red portion represents the propulsion

module.



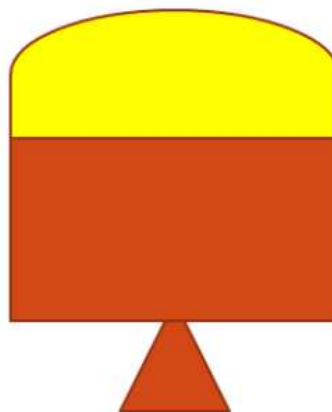
**Figure 2: Spacecraft with Separable Return Module**

### **Rendezvous Mission**

The idea in this configuration is to go and impact the object at a minimal touch down velocity and collect sample before the spacecraft bounces back. All the complexities of the previous mission exist here also. Additional constraints on the payload design due to short duration available for sample collection also exists. Hence this idea was dropped.

### **Spacecraft with Impactor**

The impactor would impact on to the target creating a debris field. The spacecraft would move through the debris field collecting samples and returning it back to earth. This is feasible within our constrain limits and is the best out of the lot. The impactor however is an additional mass being carried all the way. This is in fact a combination of the Star Dust mission and Deep Impact mission of NASA. This seemed to be feasible, but additional payload requirement for the impactor adds to the energy requirements of the mission. A glimpse of this configuration is as shown in figure 3. The yellow portion acts as the impactor, while the red portion as the propulsion cum reentry module.



**Figure 3: Spacecraft with Impactor**

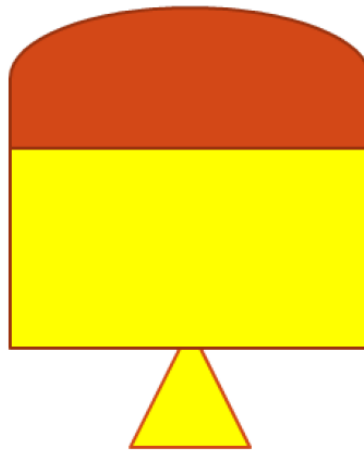
### **Finalized Configuration**

Upon rethinking, it was felt that the above configuration can be used with minimal modifications. It was proposed and accepted that the propulsion module itself may be used as the impactor. This would not only reduce the total mass of the system, but also make the mission simpler and reliable due to lesser number of separation planes. The finalized

configuration consists of the propulsion module which will propel the spacecraft from EPO to the target object. It would then separate from the reentry module and align itself to a collision trajectory with the object. Upon collision, the module is expected to create a debris field, through which the reentry module will pass through, collecting the required samples. The reentry module will then reorient itself and propel back to the return trajectory. It would park itself in an EPO from where it would be de-boosted and brought back to the earth's surface. The proposed configuration is detailed in figure 4. The yellow portion represents the propulsion module and the red portion represents the reentry module.

### Engine Design

It was decided to reduce the complexities that the same engines would be used for both the propulsion and the reentry module. The engine parameters were defined as is shown in table 1. MOC based method was used for design of the nozzle contour. The readily available LAM injector was selected as the injector as they have been proven for such applications. Figure 5 gives the output of the MOC code written in MATLAB.



**Figure 4: Finalized Configuration**

**Table 1: Finalized Configuration details**

LAM Engine Parameters	
<b>Engine Type</b>	Pressure Feed Engine
<b>Thrust</b>	400N
<b>Propellant</b>	MMH-MON3
<b>AR</b>	12( $\gamma = 1.3$ )
<b>Me</b>	6
<b>I<sub>sp</sub></b>	320(from LAM data)
<b>M</b>	1.25kg/sec

The parameters given in the above table has been arrived at, based on the previous experience and consultation with literature. The data published through SP documents from NASA have been of great help in selection of these parameters. The above nozzle configuration was arrived at using the Thrust Optimized code for minimum length nozzle using the MOC method. The profile was validated with the Rao Approximation code and was found to match within 10 percentage of deviation on the nozzle contour.

The contour was subjected to CFD analysis to ensure the efficiency and the results obtained were as per predictions. Figure 6 presented is the discretized domain using Pointwise software and Figure 7 is the representation of the CFD analysis Mach number Plot. Commercially available CFD++ software was used for this analysis The SST (Shear

Stress Transport) turbulence model, which is a widely used and robust two equation eddy-viscosity turbulence model, was used in the simulations for modeling the turbulences involved. The model combines the features of k- $\omega$  turbulence model and k- $\epsilon$  turbulence model such that the k- $\omega$  is used in the inner region of the boundary layer and is switched to the k- $\epsilon$  in the free shear flow. This model is typically used for cases involving separating flows and adverse pressure gradients. This model was selected based on the fact that the current study investigates the flow separation from the pintle and also its further interaction with the ambient conditions.

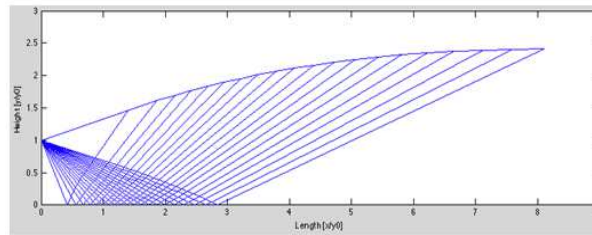


Figure 5: Nozzle Contour Design using MOC

Courant number was ramped from 1 to 50 during the course of the simulation and the numerical simulation was run at steady state up to the point where the residuals dropped down to below  $10E5$  value. The overall design of the engine is shown in figure 8

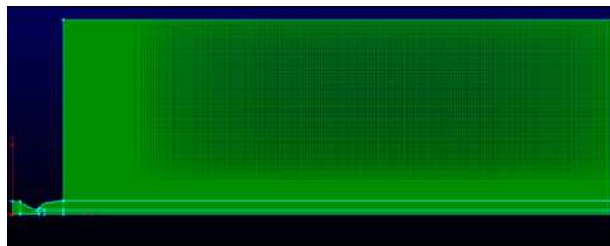


Figure 6: Nozzle Contour Design using MOC

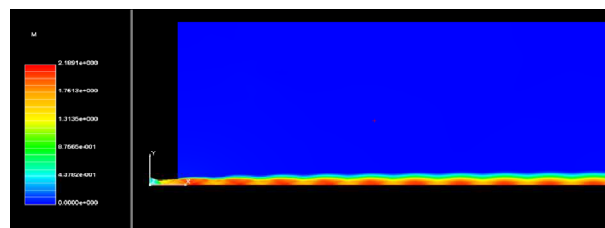


Figure 7: Nozzle Contour Design using MOC

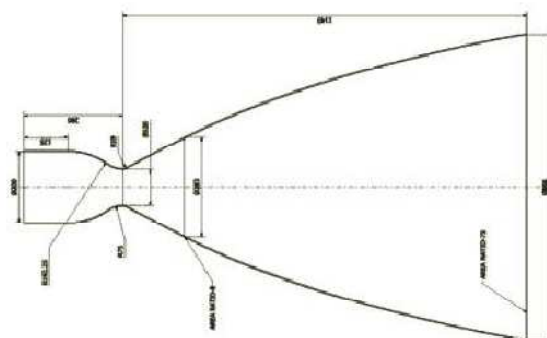


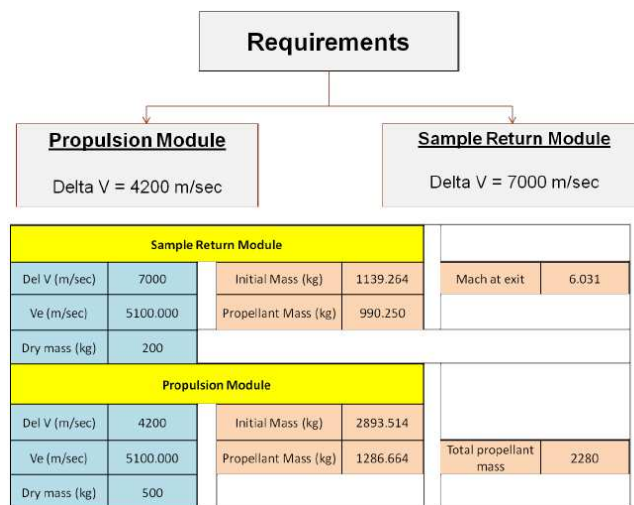
Figure 8: Nozzle Contour Design using MOC

### Propellant Quantity Estimation

As per the standard practice, the propellant quantity was estimated as below. Figure 9 shows the various parameters used and the final quantity of propellant arrived at. Simple rocket equation was used for estimation of the propellant from the delta v obtained from the trajectory team.

### Pressurization and Fuel Feed System Design

From the above analysis, the propellant quantities required were estimated for the propulsion module. Standard procedure for finding the propellant tank dimensions were followed in par with guidelines given in NASA SP125 [2] was arrived to the details presented in Figure 10.



**Figure 9: Propellant Quantity Estimates & Requirements**

Prop Mass (kg)	1290.000	MON3 (lit)	564	MON3 (kg)	812
MR	1.700	MMH (lit)	543	MMH (kg)	478
Propulsion Module		Total Mass of propellant is assumed to be 2% more for accommodating outage			
Total Fluid Mass	1294	Avg Tank Pr (bar)	18	Mass of Helium (kg)	4
Structural Mass Fraction (%)	9	MON3 tank Vol (lit)	581		
Mass of Structure (kg)	128	MMH tank Vol (lit)	559		
Mass of Stage	1422	Total Tank Vol (lit)	1140		

**Figure 10: Propulsion Module Parameters**

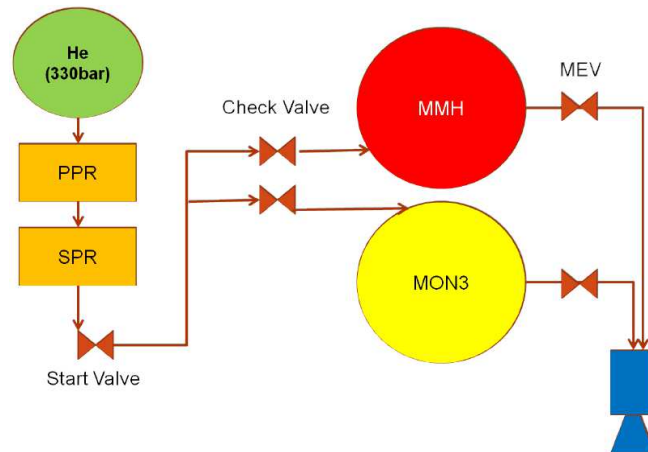
The tank volumes were kept at 1.3 times the arrived at volumes to account for the ullage gases. The ullage gas was finalized to be helium. Figure 10 also give the mass of helium to be stored at 330 bar in bottles and stepped down using regulators. Similar analysis was done for reentry module and the arrived at parameters are furnished in figure 11. It was also decided to use cold gas pressurization system using stored helium gas for both the propulsion and the reentry module [3]. It was found better to go for pressure fed system of propellant feeding for both the modules as the thrust requirement is less However in case of the propulsion module, the pressurization gas would be stored at 330 bar in the gas bottles and will be stepped down to 17 bar abs before entering the propellant tanks for pressurization. This requires active

pressurization systems. The pressurization system designed for this purpose has the configuration given in figure 12.

Prop Mass (kg)	710.000	MON3 (lit)	310	MON3 (kg)	447
MR	1.700	MMH (lit)	299	MMH (kg)	263
Sample Return Module		Total Mass of propellant is assumed to be 2% more for accomodating outage			
Total Fluid Mass	716	Avg Tank Pr (bar)	50	Mass of Helium (kg)	6
Structural Mass Fraction (%)	9	MON3 tank Vol (lit)	320		
Mass of Structure (kg)	70	MMH tank Vol (lit)	308		
Mass of Stage	783	Total Tank Vol (lit)	628		

**Figure 11: Sample Return Module Parameters**

In case of the reentry module it was found that due to space constraints, the propulsion system with regulators is not feasible. Hence blow down mode system with propellant tanks pressurized to 50 bar abs was chosen to be used.



**Figure 12: Pressurization System Configuration for Propulsion Module**

### Simulation of Pressurization System of Propulsion Module

Visual basic code was written based on the flowchart annexed in the appendix at the end of this document. The code was run to check for the adequacy of the helium filled in the gas bottles. It was found that at the end of the burn duration during reentry, the helium gas would start to act in a blow down mode. This shows that the amount of helium filled in is lesser than what is required for the normal duration throughout the mission. However, it may be noted that this happened only the end of the mission and in the worst case scenario. Even with this problem the engines would fire, but the firing duration will have to be extended owing to the lesser thrust derived from the engines fed with lesser pressure.

Detailed analysis is warranted in this aspect in case the mission is going to be accepted for realization. Figure 13 gives the GUI of the code written in this regard and figure 14 gives the validation of the results obtained from the code for PSLV C14 PS2 stage tank pressure data.



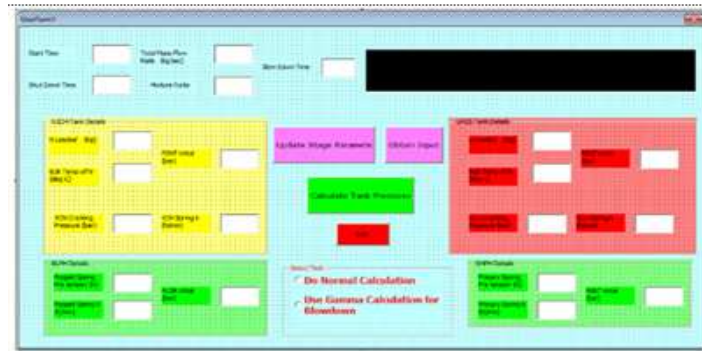


Figure 13: GUI of Visual Basic Code

### Overall Configuration and Line Routing of the Propulsion Module

The propulsion module with all hardware placed and with tentative line routing is as shown in figure 15 & figure 16. This is a tentative scheme and may be subjected to changes depending on the final CG locations and its requirements.

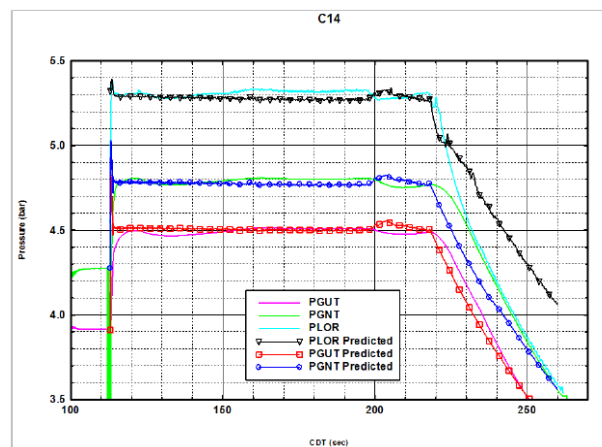


Figure 14: Validation of Visual Basic Code

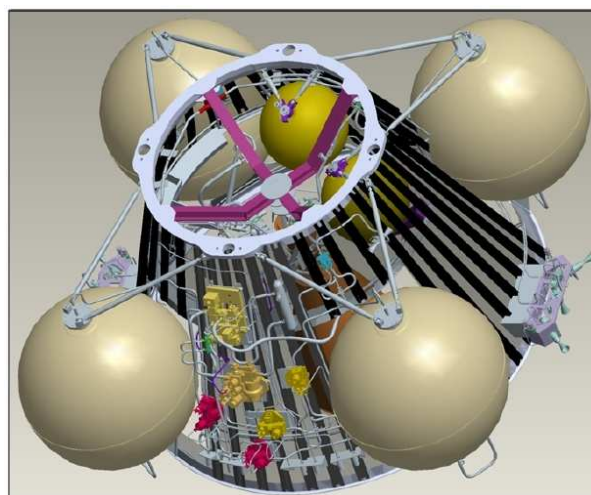
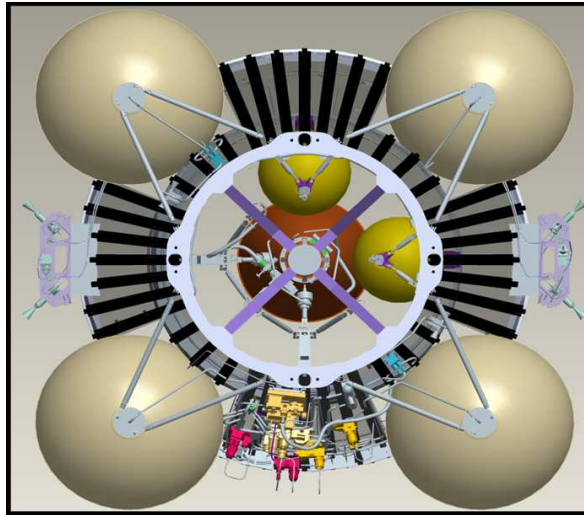


Figure 15: Pressurization System Configuration for Propulsion Module Isometric View





**Figure 16: Pressurization System Configuration for Propulsion Module Top View**

## CONCLUSIONS

The design of the propulsion module for '2010TK7 sample return mission' is finalized. Preliminary design and analyses of propulsion module line routing can be defined based on the analysis. Sufficient margins available for the analyzed structures. The code verification allows the user for further design aspects. More optimization studies, interface design and assembly details are to be worked out in the detailed design phase. A structural qualification and testing plan is to be generated after detailed design.

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